Look, Ma, No HANS![1]

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Abstract—As part of the low cost design, the spinning Genesis spacecraft has no High Accuracy Navigation Systems (HANS) on board, such as gyros or accelerometers. All science requirements are met, although accurate pointing estimation is limited to spin rates less than 2 rpm, due to star tracker characteristics, and less than 28° off sun, because of two-axis sun sensor characteristics. Payload contamination concerns result in thruster locations that produce uncoupled forces, and significant translational delta-V, whenever pointing or spin rate are changed. Nevertheless, with creative systems, mission and subsystems design, planning and operations, Genesis should exceed all science objectives. The Genesis lessons learned will benefit future mission design and operations in the areas of both planned activities and off-nominal situations.

TABLE OF CONTENTS

1. INTRODUCTION

Genesis is part of the Discovery class missions: Faster, Better, Cheaper. The 636 kg (1400 lb) spacecraft was launched August 8, 2001, on a Delta 7326 from Cape Canaveral. It is currently in orbit around the L1 Libration point between the Earth and Sun. Genesis will collect charged particles across selected energy regions from the solar wind for more than 2 years. The capsule containing the solar wind samples will return to Earth on September 8, 2004 after a spectacular mid-air capture over northwestern Utah in the Utah Test and Training Range (UTTR). Analysis of the samples will ultimately provide insight into the primeval solar nebula and evolution of the solar system. A mission overview is shown in Figure 1.

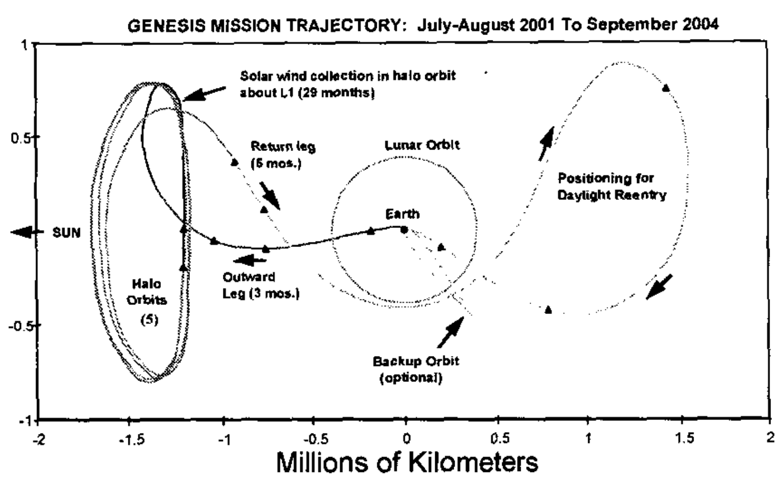

Figure 1. Mission Overview

[1] 0-7803-7231-X/01/$10.00©2002 IEEE

The following discussion will show the progression of a proposed concept into an operational mission. Along the way, some anticipated obstacles disappeared, only to be replaced by other unexpected issues. The design and operational team have met these challenges by developing hardware and software workarounds, and drawing from a bag of tricks normally reserved for operational contingencies. Yet, the mission still has more than adequate performance margin, as exemplified by actual flight performance to date.

2. GENESIS, A DISCOVERY CLASS MISSION

The Prime Directive of a Discovery class mission is the Faster, Better, Cheaper (FBC) paradigm. 'Faster' means compressed schedules from concept development to launch. 'Better' means capturing the imagination of both the public and scientific communities, as well as maximizing mission success. 'Cheaper' means, of course, low cost for the entire project cycle: Design, development, test and integration, launch and operations.

Embedded in this approach is the opportunity to offer greater science return with an increased number and diversity of missions for the same dollars, to provide broader public appeal with more frequent missions, and to avoid placing all of the science eggs in a single mission basket. If something goes wrong, degrading or losing one small mission is much better than losing a 'mega' mission.

Genesis was proposed as a Discovery mission called Seuss-Urey in 1995 . It was selected as a Discovery mission in late 1997 with a launch anticipated in December, 2000.

Genesis' mission is to collect constituents of the solar wind that can be used to determine the elemental composition of the original solar nebula. To do this, Genesis escapes the Earth's magnetosphere using a Delta 2 launch vehicle and orbits around the sun-Earth L1 libration point, collecting solar wind particles [1]. After a minimum of 2 years, a capsule containing the collected particles returns to Earth. Because the collection material is fragile, a parachute descent with a mid-air capture by helicopter is employed. To support the use of manned aircraft for sample recovery requires a return to Earth in daylight. Figure 2 illustrates the launch, cruise (lower deck), science (upper deck) and mid-air Sample Return Capsule (SRC) recovery configurations.

To minimize risk, and maximize mission success, the spacecraft design strategy incorporated many heritage components and processes, especially from the Stardust spacecraft. Simplification was applied as much as possible, such as spin rather than 3 axis stabilization and star trackers without an Inertial Measurement Unit (IMU).

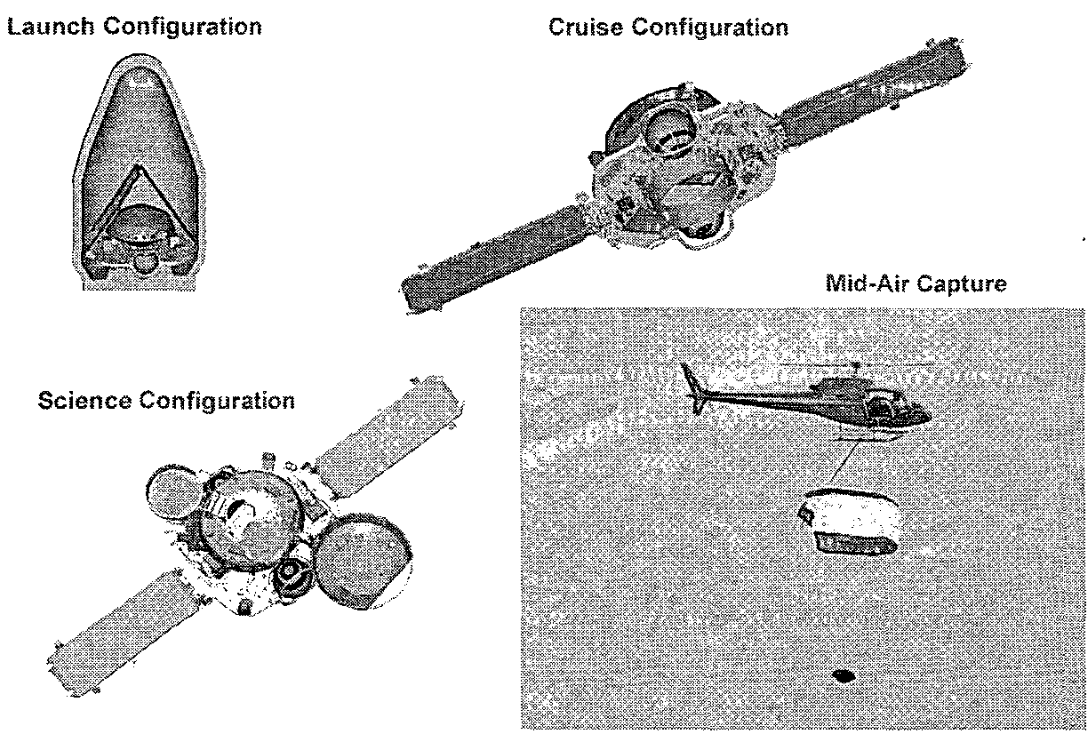

Figure 2. Mission Configurations

System reliability was enhanced through single fault tolerance, including redundancy in critical components, and having a hierarchical fault protection system that was integrated into the spacecraft design.

The failures of Mars Climate Orbiter (MCO) in September, 1998, and Mars Polar Lander (MPL) in December, 1998, impacted Genesis development in several areas. The Genesis launch was delayed six months until the July-August 2001 opportunity in order to avoid potential conflicts with the Mars Odyssey launch period in early April 2001. The subsequent mission redesign included a fifth halo loop around L1, and an extra six months of science collection time. The launch delay provided time for additional analyses and testing, especially in the areas of contingencies and risk reduction. Another mission enhancement involved more communications and integration amongst spacecraft design/development/test, mission design/navigation and mission operations teams.

3. SCIENCE AND PAYLOAD

The solar wind holds valuable clues to the isotopic composition of the sun, similar to a fossil record. Genesis will examine mostly nuclei or ions, and electrons ejected by the solar corona. The solar wind particles of interest fall into three basic regimes: "Fast" particles from coronal holes, "Slow" or interstream particles that come from coronal boundary regions, and Coronal Mass Ejections (CME) which are transient explosive events. Carbon, Nitrogen and Oxygen are some of the elements of interest, as are the noble gases and a number of rare elements. Genesis hopes to measure abundances of most of the elements in the periodic table.

Genesis has two monitors located on the main bus, an ion monitor (GIM) and an electron monitor (GEM). As the spacecraft spins at 1.6 rpm, the GEM and GIM sensor fields of view sweep across space to measure energy levels. A science algorithm in the flight software uses the monitor data to determine which solar wind regime is prevalent.

Spin axis (+x) pointing must be maintained to within 2 degrees of the solar wind direction, which is about 4.5 degrees off of the Earth-Sun line. Pointing accuracy includes wobble, caused by principle axis misalignment, and nutation, caused by external or internal torques. Pointing is adjusted by daily precession maneuvers, at about 1 degree per day. Downlink telemetry is used to reconstruct the pointing history, and to correlate it with the sensed GIM/GEM data by ground analyses.

The part of the science payload that collects solar wind particles is within a canister inside of the SRC. The canister contains a stack of four collector arrays, 3 for each of the solar wind regimes and one bulk collector. An additional bulk collector is in the canister cover. When stowed, the collector stack covers an ion concentrator that focuses incoming ions using electromagnetic grids.

A few months after launch, after the SRC has been opened and outgassing has been completed, the canister lid is opened, and the four collector arrays are moved from their stowed positions, directly above the concentrator, to their deployed stacked positions, exposing the concentrator. When the science algorithm determines what solar wind regime is active, then the appropriate collector array is moved into its 'unshaded' or collection position. The concentrator voltage is also adjusted as needed. When a new wind regime is detected, the old regime collector is returned to its 'shaded' position in the stack, and the new collector is unshaded.

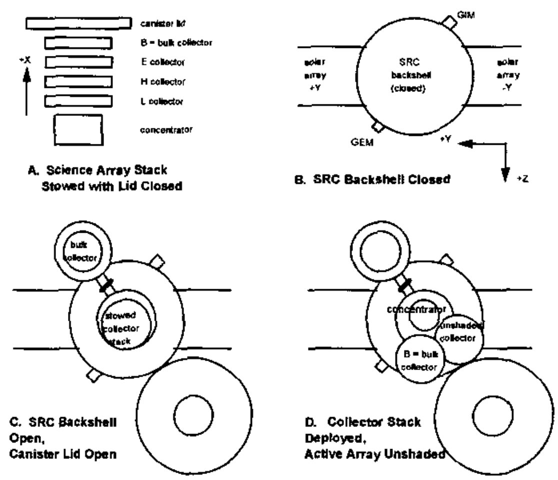

Figure 3. Payload Canister and Collector Array Sequence

This autonomous operation continues for five loops around L1, or about 2 1/2 years. The only interruption to these operations is when the spacecraft pointing, spin rate or trajectory is changed using 0.9 N (0.2 lbf) thrusters. Mechanism motion, whether it is the SRC back shell, canister lid or collector arrays, will introduce nutation, just as thruster operations also induce nutation. To prevent adverse dynamic interaction between thruster and mechanism operations, only one kind of activity is allowed at any given time. The activity sequence includes a suitable time period for nutation damping after the activity.

Preventing contamination of the ultra-pure materials in the collector arrays (silicon wafers) and concentrator (silicon and diamond tiles) is very important. Impurities may be introduced through surface contamination, especially from thruster exhaust. To prevent this, spacecraft thrusters are located on the back side of the bus, pointing away from the payload side. Although a pure form of hydrazine fuel is used, contamination analyses require that no more than 30 kg of fuel be cumulatively expended during science operations when the canister is open.

4. Spacecraft Design

4.1 Spacecraft Description

The Genesis spacecraft was developed based on hardware and software successfully flown on other missions. The spacecraft was designed to be a sun-pointing major axis spinner, consisting of an equipment deck and a sample return capsule (SRC). The equipment deck provides primary structural support for the spacecraft subsystems, the Ion and Electron monitors and the sample return capsule. The SRC contains the science canister solar wind collection system, including the electrostatic concentrator and passive collector arrays.

Figure 4 shows the overall spacecraft with sample return capsule attached to the upper or topside deck. Two solar arrays provide power while sun pointing. A rechargeable battery provides power during maneuvers, when the spacecraft is pointed away from the sun. Redundant low gain antennas are mounted to the solar arrays. Redundant transponders are used to support communications with the Deep Space Network (DSN) stations. Two propellant tanks feed redundant hydrazine thrusters. Redundant star trackers, combined with solar array mounted two axis sun sensors and deck mounted sun sensors, provide attitude determination.

Figure 4 also highlights the lower deck, or underside, of the spacecraft. The secondary battery is located inside the launch vehicle adapter ring. The medium gain antenna provides high data rate communications with the DSN during science collection. Attitude control is provided through eight small, called RCS (Reaction Control System), thrusters canted from the spin axis. These thrusters are used to control the spin rate, precess the spin axis and perform small delta-V maneuvers. Large delta V maneuvers are performed with four large, called TCM (Trajectory Correction Maneuver), thrusters axially directed.

4.2 Subsystem Description

MECHANICAL - The primary structural component is the equipment deck that houses most of the spacecraft components, either on the forward (+x-axis, upper, sun facing) deck or on the aft (-x-axis, lower, Earth facing) deck. The SRC is mounted on the forward deck, and the launch vehicle payload adapter is mounted on the aft deck.

Spacecraft mechanisms include the SRC hinge and retraction device, the SRC separation spring devices, solar array hinges and dampers, solar array mounts for the sun sensors and low gain antennas, and various retention and release (R&R) devices that allow the solar arrays to deploy and the SRC to be released. There are also SRC mechanisms associated with the Detachable Aft Conic Section (DACS) release and SRC latches.

THERMAL - Thermal control relies on a passive design, supplemented with autonomous and ground controlled heaters. Radiators on the aft (-x-axis) side of the spacecraft deck provide heat shedding for major heat sources (power control assembly, radio transponder, and command and data handling subsystem). Surface coatings and multi-layer insulation (MLI) are also a key part of thermal design.

POWER - The Electrical Power Subsystem (EPS) provides the energy collection, storage and power distribution for the spacecraft. The subsystem consists of four major components: two solar array wings populated with silicon cells, rechargable battery, Power Controller Assembly (PCA), and Pyro Initiation Unit (PIU). The EPS is a direct energy transfer design, where battery voltage equals bus voltage. The battery is on trickle charge for most of the mission, except for maneuvers greater than about 45° off sun.

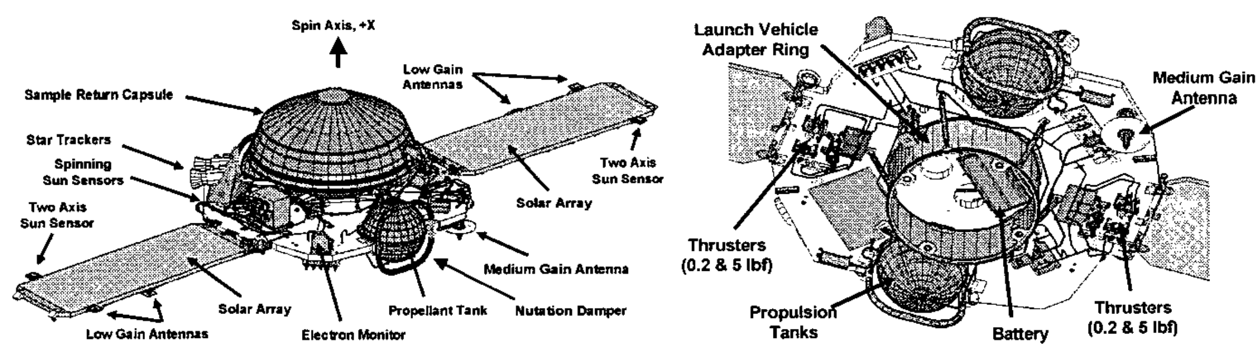

Figure 4. Spacecraft Configuration: Upper and Lower Decks

The two solar arrays are capable of generating 281 watts at 1.012 A.U., 10° off-sun, 65°C at end of life with losses due to radiation, UV contamination and micrometeoroids. The battery has a nameplate capability of 16 amp-hr at 28 volts. Minimum allowable bus voltage is 24 volts. The battery provides a total capacity of 448 w-hr @ 100% state of charge (SOC), and 313 w-hr at 70% SOC.

The PCA supports energy collection, storage and distribution functions, and provides the telemetry interface to the command and data handling subsystem. It also provides shunt and battery logic, and power distribution via load switching, memory cards, and the Motor Actuation Driver (MAD) card.

The PIU provides the high power interface to the propulsion subsystem and redundant pyrotechnic devices. For propulsion, power is delivered to latch valves and thrusters. For pyrotechnics, power is delivered for solar array deployment, GEM and GIM cover deployment, and SRC pyros (cable cutters, hinge and spacecraft bus separation springs).

TELECOMMUNICATIONS - The spacecraft's communications subsystem has two transponders (primary and redundant), four low gain patch antennas (LGA), two facing forward (+x-axis) and two facing aft (-x-axis), and a medium gain antenna (MGA) facing aft. All antennae are capable of both transmitting (Tx) and receiving (Rc).

The spacecraft's four LGA antennas are located on the solar arrays. The LGA field of view (FOV) is ±60° about the x-axis (+x-axis for forward LGAs, -x-axis for aft LGAs). The MGA is located on the +z, -y-axis corner of the aft deck.

Upon transmission to Earth, the data are received by either 26-m or 34-m DSN antennas. There are 8 anticipated downlink data rates, from 1050 bps to 47400 bps, depending on spacecraft antenna, DSN antenna, and mission phase. There are fixed data rates for uplink and for safe mode downlink.

ATTITUDE CONTROL - The attitude control subsystem (ACS) provides spin attitude and rate knowledge, and open loop thruster control. Attitude sensors and estimation algorithms are used to provide knowledge that is then used by thruster control algorithms to change spin rate, precess the spacecraft angular momentum vector, provide trajectory corrections through delta-V implementation, and for contingency operations.

There are three attitude sensor types: digital 2-axis sun sensor (DSS), spinning sun sensor (SSS) and star tracker (ST). Redundancy and cross-strapping are used to assure reliability.

Each star tracker has a field of view of 33.5° x 23.5°. Star images are captured on a Charge Coupled Device (CCD). Although the star tracker has the capability of outputting a direction quaternion, this is not used for on-board attitude estimation (AE). CCD calibration and star tracker checkout are performed on an as needed basis.

Primary attitude knowledge is provided by an attitude control algorithm called Spin Track. Spin Track uses individual star measurements from the star tracker, and a forward-facing sun sensor to determine a given star's brightness, spin-phase angle, and off-sun angle. These data are passed to a C&DH resident flight software algorithm, which then compares the measurements to an on-board star catalog for star identification. Attitude solutions can be calculated from star vectors and the sun vector. Spin track will estimate the spin axis (angular momentum or H) vector, spin rate and principal axis misalignment. Spin track usage is effective when the spacecraft +x axis is within 28° of the sun (DSS limit), and spin rates < 2 rpm (star tracker limit). The accuracy of this process is dependent on nutation levels.

The spinning sun sensor measures the sun crossing angle. The digital sun sensor is a two-axis version of the SSS, but mounted with a view along the spin axis. SSS and DSS processed output provides spin axis off-sun angle and spin rate. Under high nutation and/or sun proximity to the x-axis, multiple sun crossings can occur at irregular time intervals. These can severely degrade knowledge of the spin rate. Sun pointing keep out zones (KOZs) were established to avoid this occurrence. Mission flight rules, and both flight and ground software, enforce the keep out zones.

ACS is also responsible for the passive nutation damper system. This system consists of a viscous fluid filled tube that surrounds each propulsion tank and fluid control system. The shorter the nutation time constant, the quicker nutation is damped out. In general, the time constants diminish as fuel is expended and/or spin rate is increased. A typical time constant is 2.5 hours when the spacecraft has half its fuel left, in the science configuration, and the nominal 1.6 rpm spin rate.

PROPULSION - The propulsion subsystem includes two propellant tanks that hold at least 71 kilograms of usable hydrazine each, two pressurant bottles for fuel pressurization, six Rocket Engine Modules (REMs) that hold a total of 12 thrusters, latch valves and all the fluid lines that connect the propulsion components. Four REMs hold a pair of 'RCS', or 0.9 N (0.2 lbf), thrusters each and the remaining two REMs hold a pair of 'TCM', or 22 N (5 lbf), thrusters each. The REMs are configured to provide a fully redundant system. They are located on the aft deck to prevent contamination of the science payload.

During thruster operations, each active thruster is fed by hydrazine fuel from the two tanks. When thrust is needed, a propulsion valve is opened to allow hydrazine to flow across a hot catalyst bed. The 'cat' bed transforms the liquid hydrazine into heated gas that is expelled out the nozzle, producing the desired thrusting effect.

The fuel tanks are fully pressurized at launch. Over the mission life, as fuel is consumed, the blowdown propulsion system will result in reduced end of life pressures. Thrust levels may drop by over a factor of four from the beginning to the end of the mission. The specific impulse (Isp) of the thrusters also has diminished performance as tank pressure drops.

Small maneuvers, using RCS thrusters, are typically limited to 2.5 m/s due to thruster thermal and off sun maneuver time limitations, but can be higher for a sunward maneuver that allows for multiple 2.5 m/s segments. In-flight experience may change this limit to allow larger maneuvers to be performed. On the other end of the performance spectrum, in-flight experience may also prove it necessary, toward the end the mission, to use the TCM thrusters to perform attitude precessions. Large delta-V maneuvers use TCM thrusters.

COMMAND AND DATA HANDLING AND FLIGHT SOFTWARE - The Command and Data Handling (C&DH) subsystem is housed in a box on the spacecraft's forward deck. The C&DH provides time definition and command and data interfaces with all other subsystems. It is fully redundant and single fault tolerant. The C&DH contains multiple processor and memory cards and VME buses. Some of the principle cards are the Flight Processors that contain the central processor unit, dynamic random access memory (DRAM), two different backup memory devices, the Payload and Pointing Interface Card (PPIC), and the Command Module Interface Card (CMIC).

Flight Software (FSW) includes the onboard code which runs the spacecraft, including fault protection. FSW has a selectable operating speed and utilizes less than 60% of processor capability.

DRAM, 128 MB (mega bytes, 1024x1024x8 bits) worth, is used for all operations. FSW has been allocated 32 MB, and telemetry storage has been allocated 96 MB for science and engineering data. This is sufficient for multiple playbacks in the current downlink strategy.

Key FSW functions include the PPIC driver (hardware serial interfaces, discrete and analog input/output), telecommunications, EPS, structures and mechanisms, fault protection, ACS and payload operations.

Fault protection is responsible for failure detection, response and recovery. A hierarchical detection strategy isolates the failure. Responses include switching from primary to redundant strings and/or swapping C&DH sides. If needed, an autonomous safe mode is entered which reconfigures the vehicle to minimize electrical power loads, continues fault detection and response, and (if the spacecraft is > 35° off sun) precesses quickly to a sun pointed attitude for solar array power.

SAMPLE RETURN CAPSULE - The Sample Return Capsule (SRC) has four major elements: structure, thermal, avionics and payload canister. The SRC structure is composed of the heat shield half and the backshell half. The heat shield portion includes redundant batteries, avionics boxes, Global Positioning Satellite (GPS) receiver, and entry thermal protection. The backshell portion includes the Detachable Aft Conic Section (DACS) which contains the descent parachute. There are also mechanical devices such as the hinge, which opens and closes the backshell, four latches, and several pyrotechnic devices. Thermal components of the SRC include the battery radiator, MLI, surface coatings, and all the insulation to protect the SRC from entry heat loads. Avionics include a patch antenna, VHF locator beacon, GPS receiver, UHF transceiver, battery, and supporting electronics.

4.3 Design Decisions

During the design process, many choices and decisions were made in selecting components, processes, testing, etc. Some of the key decisions are shown in Table 1, roughly in order of occurrence. Selection rationale was the result of trade studies, with a very important aspect being their mission impact. Some of the early decisions had a greater impact than originally envisioned, in terms of design and operational complexity. However, it should be noted that Genesis has met <u>all</u> science and mission requirements, including more than adequate performance margins and reserves.

Most of the major impacts involved Systems, ACS and MDNav (Mission Design and Navigation). Resultant actions included:

- Detailed maneuver timelines were developed for all major activities, especially those requiring off sun precessions. Electrical power balance, nutation levels, precession strategy, etc. were evaluated, and the activities were adjusted as needed. These timelines had the benefit of directly supporting command block generation.
- The regime of sunward and anti-sunward keep out zones were defined for all mission circumstances. These were required to support operations, such as flight rules and constraints, ground software and flight software capabilities.
- Maneuver decomposition techniques (discussed later) were developed to understand what kinds of maneuvers would be required, and to support the operational maneuver design process.

Table 1. Key Design Choices

Design Choice	Rationale	Mission Impact
Mid-air SRC capture	Protect sample collection Entry similar to Stardust Robust capture technology	Parafoil and navigation aids needed Helicopter capture training Return in daylight
Single S/C battery	Cell out redundancy Mass margin Simpler, and lower cost	Nominal off sun time limited to about 85 minutes Contingency off sun time between 40 and 130 minutes Integrate battery SOC with fault protection strategy
Lower deck thrusters	Avoid science contamination Heritage avionics limitations Simpler, and lower cost	Uncoupled forces produce ΔV during all maneuver activities Maneuver decomposition required in maneuver design process Small forces complicate Orbit Determination
No gyros	No firm requirement Saturation with spinning S/C	Attitude estimation (AE) uses celestial sensors
No accelerometers	Not necessary for most maneuvers (low accuracy) Insufficient closed loop accuracy Not cost effective	Maneuver open loop rather than closed loop - ΔV cutoff based upon burn duration - Spin rate adjustment and precession use other sensors ΔV assessment relies on Navigation reconstruction Small forces and ΔV calibrations needed
Replace heritage star tracker with 'new'	Heritage tracker needed spin rates <0.03 rpm for star identification New tracker had internal AE	New tracker valid for spin rates < 2 rpm
Use star tracker and DSS in Spin Track AE	Spin Track robustness and accuracy for science collection	Spin Track valid for spin rates < 2 rpm (tracker limit) and < 28° off sun (DSS limit) Use SSS when spin track not available
Use SSS for AE	Available when Spin Track not useable Contingency Attitude Estimation	Estimates only planar (cone angle) pointing direction Measurement ambiguities when pointing close to, or far from, sun - KOZs required for near sun and anti-sun directions - ACS FSW complicated by KOZs - Maneuver design and execution more involved

- Maneuver types and timelines were defined by ACS, consistent with S/C and mission constraints. These maneuver types are a function of S/C configuration, range of $\Delta\underline{V}$ magnitudes and range of $\Delta\underline{V}$ off sun angles. The maneuver types define the sequence of spin rate control, precession and $\Delta\underline{V}$ burns that are needed for maneuvering from a nominal pointing and spin rate to a commanded pointing, such as a trajectory correction $\Delta\underline{V}$, and back to nominal. Implicit in the sequence is recognition of thruster, KOZ and attitude estimation (AE) constraints.

Maneuver decomposition computes the component $\Delta\underline{V}$ activities, magnitude and direction, of all precessions, spin rate changes and the commanded burn $\Delta\underline{Vc}$, which will combine to produce the desired translational $\Delta\underline{Vd}$, magnitude and direction, or

$$\Delta\underline{Vd} = \sum \text{Precession } \Delta\underline{V} + \sum \text{Spin rate change } \Delta\underline{V} + \Delta\underline{Vc}$$

The desired $\Delta\underline{Vd}$ is determined by the Navigation team. Commanded $\Delta\underline{Vc}$, spin rate changes and precessions, are determined by the spacecraft (ACS) team using decomposition software.

There are two broad categories of trajectory correction maneuvers: single and double legs. Single leg maneuvers, simply put, precess to a burn direction, perform the $\Delta\underline{V}$ burn, and return to the nominal pointing conditions. There are spin rate changes prior to and sometimes after the $\Delta\underline{V}$ burn. If the maneuver decomposition process determines that a single leg maneuver $\Delta\underline{V}$c falls into a keep out zone, then a double leg maneuver is required. The double leg maneuver, sometimes referred to as a dogleg, essentially performs two single leg maneuvers that are symmetric about the sunline, and also do not violate the KOZs. The difference between legs is in the magnitude of the $\Delta\underline{V}$c for each leg.

Figure 5 is a simple planar representation of the maneuver activities. DVd is the desired $\Delta\underline{Vd}$ magnitude and θ is the off sun angle of $\Delta\underline{Vd}$. DVp represents the $\Delta\underline{V}$ from precession maneuvers (spin rate change $\Delta\underline{V}$ is not shown). DVc is the commanded $\Delta\underline{Vc}$ magnitude and ψ is the off sun angle of $\Delta\underline{Vc}$. For single leg maneuvers, DVc is always smaller than DVd and ψ is always larger than θ. For double leg maneuvers, only the first leg is shown and only the net precession DVp is shown. DVc lies on the anti-sun keep out zone boundary.

Figure 6 illustrates typical boundaries between single and double leg maneuvers. For double leg maneuvers, there are anti-sun and near sun varieties. Trajectory biasing was introduced into the reference trajectory to minimize the possibility of having to perform the more complex anti-sun double leg maneuvers.

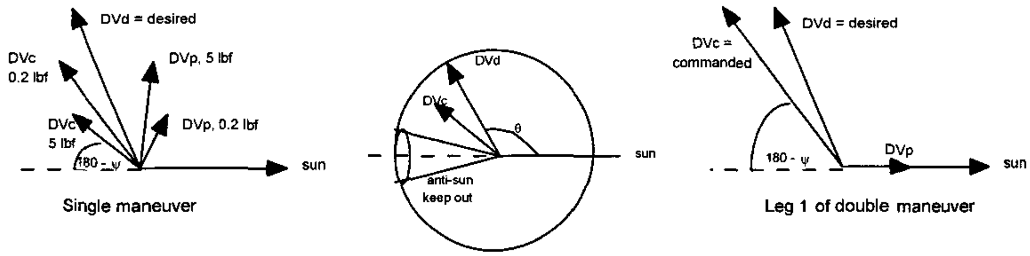

Figure 5. Planar Representation of Maneuver Activities

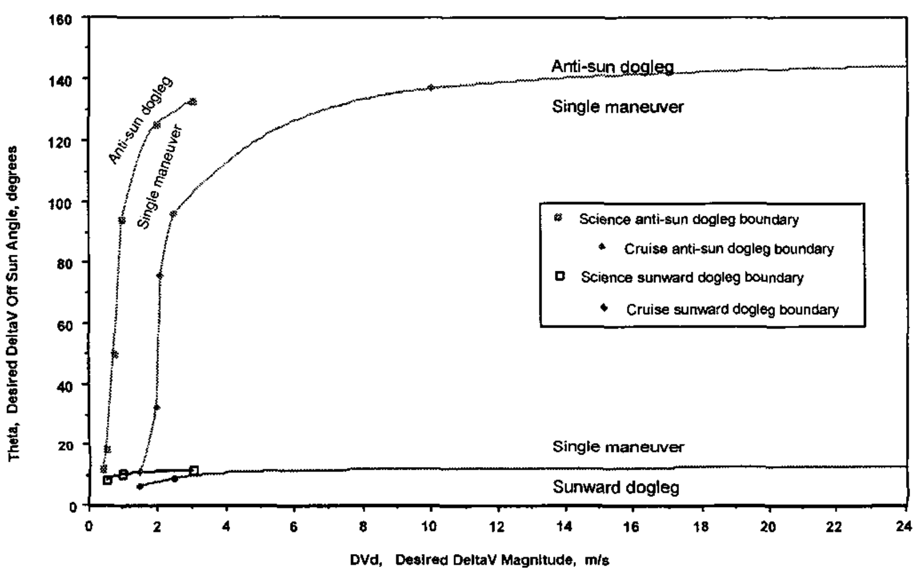

Figure 6. Single Vs. Double Leg Maneuver Boundaries

The final trajectory correction, prior to Earth entry, makes use of spin rate changes and uncoupled thrusters to create a ΔV of about 1 m/s. ΔV execution accuracy is ensured by prior calibrations that measure spin rate change vs. reconstructed ΔV [2].

5. Mission Design and Navigation

The Genesis trajectory design was accomplished using techniques innovated over the last five years that involve the application of dynamical systems theory to the multi-body problem. The technique was principally developed, and has been described in greater detail, by Lo, Howell, Barden, et al, [3,4,5,6] including specific applications to the Genesis mission.

The key aspect of the theory that enables the Genesis mission is the concept of families of trajectories, or manifolds, characterized by equipotential surfaces that circulate around the sun-earth Lagrangian or libration points L1 and L2, as well as between them. The theory incorporates a basic characteristic of these surfaces, embodied in the observation that there are stable and unstable manifolds. Trajectories that naturally wind onto libration points occupy stable manifolds. Those that wind away are contained on unstable manifolds, as in Figures 7 and 8. Crucially, there exist connections between manifolds that provide a constant energy link between L1 and L2. These are heteroclinic connections. So-called homoclinic connections also exist. They are manifolds that leave a libration point eventually to return to that point. Heteroclinic dynamics plays a central role in deterministic chaos of a dynamical system. In many ways, the Genesis mission design represents the practical application of chaos theory in nonlinear system dynamics.

In Figure 7, the manifolds leading to L1 can be seen emanating from the proximity of earth. These manifolds are used to design the transfer trajectory from launch to the vicinity of the libration point. Once the halo or, more correctly, Lissajous orbit at L1 is achieved, the mission shifts from a stable to an unstable manifold in order to facilitate the return to earth, as seen in Figure 8.

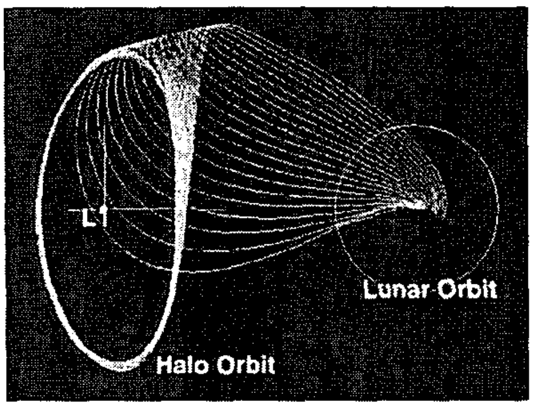

Figure 7. Stable Manifolds Winding from Earth onto L1

The Genesis trajectory utilizes a combination of stable and unstable manifolds, with an assist from the moon, to accomplish its objective of loitering near L1 for 2 1/2 years and returning to earth. In the process, remarkably, a single deterministic maneuver is required over the entire mission. Having made this claim, a one-maneuver mission is, in Genesis's case, actually only a theoretical possibility. Pragmatic necessity required the insertion of deterministic maneuvers at points along the trajectory in order to accommodate operational constraints on the spacecraft. Nevertheless, the basic design begins with the single maneuver concept, and is later modified to address the practicalities of dealing with a real spacecraft.

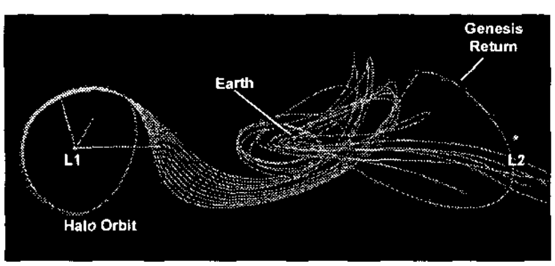

Figure 8. Unstable Manifolds and Heteroclinic Connection from L1 to L2

As it happens, the single deterministic maneuver connects the stable manifold transfer portion of the trajectory, from Earth to L1, to the return trajectory, which takes an unstable manifold away from L1, along the heteroclinic connection that loops by L2, finally returning to Earth. The critical maneuver is the Lissajous Orbit Insertion maneuver (LOI). This sets up the cycle of five loops around L1 during which most of the solar wind samples are collected, followed by the subsequent return to Earth. Thus, after execution of LOI, the spacecraft is essentially on a "free return" trajectory.

In Figure 9 we see the resulting mission design in three-view perspective, as shown in a coordinate system rotating with the earth in its orbit around the sun, the x-axis of that system being always sun-pointed.

Early in the proposal stage, a decision was made to fix the LOI point in inertial space throughout the approximately two-week launch period in order to simplify the design. This incurred a cost in ΔV since ideally the optimization could have incorporated optimizing the LOI location. On the other hand, the cost was modest and still permitted a reasonable launch period within the ΔV budget allowance, while still accommodating all the other mission DV requirements comfortably. Fixing LOI inertially had the added benefit of making the L1 "tour" and return to earth phases independent of launch date, greatly simplifying operational planning.

The launch period is governed by the ΔV required at LOI and the position of the moon. Fixing the LOI point means that there will be a particular launch date when LOI is minimum, and the LOI cost will resemble a bathtub curve with that optimal LOI date at the bottom of the curve. The moon, however, will perturb this, creating a pronounced and very abrupt kink in the midst of the "bathtub", as shown in Figure 10.

The desire was to avoid the kinks, and bias the beginning of the launch period well away from times when the lunar effect was significant. The moon's effect on the ΔV at LOI was not severe in itself, at least deterministically. The problem was that it could have a quite dramatic statistical effect when launch vehicle injection uncertainties were included. The consequence of being too close to the moon at launch, combined with worst-case launch injection errors, could conceivably require very large expenditures of propellant very early in the mission. Thus, the importance of lunar phase in determining the launch period. "Launch moon" for Genesis ran from roughly first to last quarter, or about two weeks. For our August opportunity, the launch period opened on July 30 and ran until August 14.

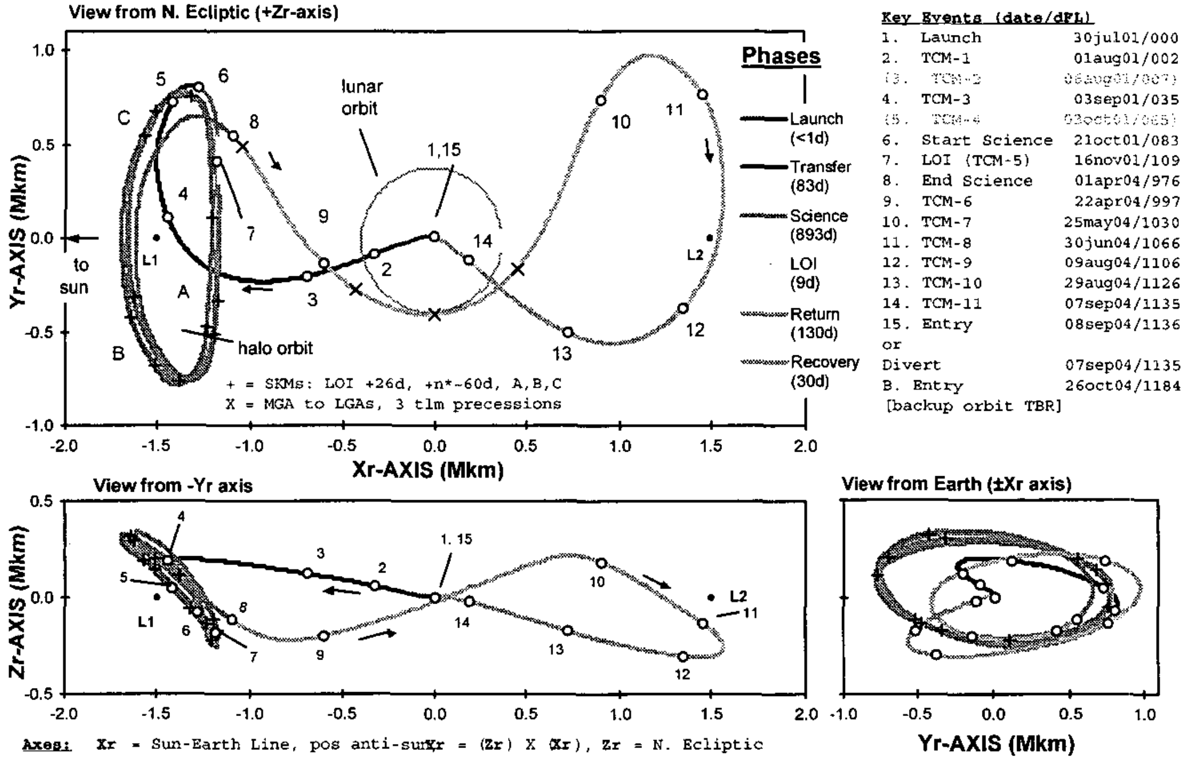

Figure 9. Genesis Trajectory in Sun-Earth Rotating Coordinates

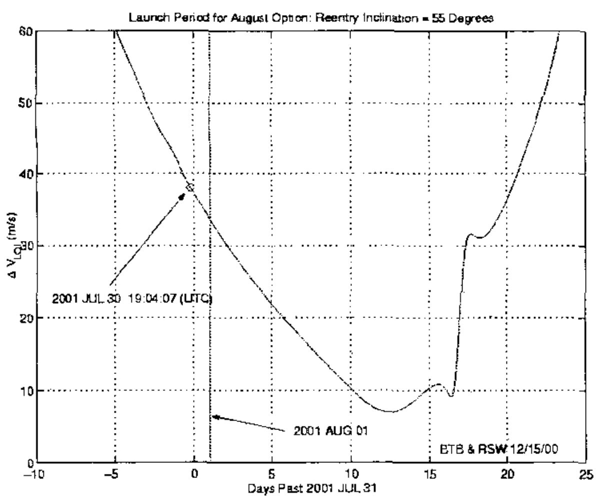

Figure 10. Launch Period Analysis – LOI ΔV vs. Launch Date

Though lunar flyby is expressly avoided, particularly for the transfer phase, the moon does exert an influence during the return. The L2 loop that precedes entry is necessary to achieve an earth approach geometry that leads to a daylight entry. But the heteroclinic L1-L2 connection is not sufficient of itself to achieve a close earth approach that leads to entry without expending ΔV. The moon provides the necessary assist, saving propellant and allowing for a trajectory design that is already elegant in its use of natural forces to achieve its final goal of returning samples to the earth by using those forces once more to complete the mission.

Figure 11 shows how the moon leads the spacecraft during the earth flyby. Though never any closer than 250 million km, the moon perturbs the trajectory enough to facilitate the final earth entry.

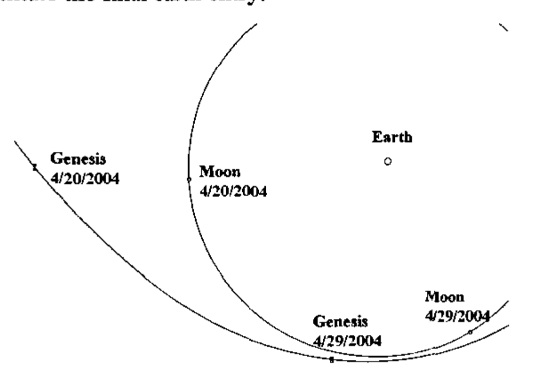

Figure 11. Lunar Geometry during Return Phase

5.1 Trajectory Design Process

Table 2 contains the primary requirements on the design of the Genesis mission.

The overriding requirement on the Genesis trajectory is to return to Earth. This also distinguishes Genesis from previous libration point missions, which adhere less closely to the reference orbit in order to remain in the vicinity of the libration point [7]. The Genesis trajectory must be navigated within fairly tight margins relative to the reference trajectory in order to assure return.

Table 2. Design Requirements Affecting the Genesis Trajectory

Requirement	Compliance
13 day launch period	16 days (July 30-August 14)
Sample collection outside magnetosphere for duration of ≥ 22 months	39 months at L1, though actual time available for sample collection is 27 months
Daylight return to UTTR from May through October	Primary return Sept 8, 2004/Backup October 24 at about 9 a.m. local
Two entry opportunities within 30 days	Backup orbit with 24 day period
Disposal of non-SRC components in compliance with NASA safety requirements	Deboost (and burn up)of bus over Pacific; Contingency deboost of spacecraft in safe mode
95% $\Delta V \leq 480$ m/s	95% $\Delta V \leq 450$ m/s
Solar occultations < 80 min	No post-separation occultations

The requirement to return also influences what time of year Genesis can be launched. Once again Genesis differs from previous libration point missions because it cannot be launched year-round. Returning to Utah during a period of favorable weather to enable helicopter recovery, combined with the need to accumulate the solar wind samples over a minimum period of time during halo cycles that are approximately six months each, determines when the mission can begin.

Genesis may be launched during two "seasons" of roughly three months duration each. The winter launch season extends from December through February. The summer season includes June through August [8]. The winter missions can accommodate 4 loops at L1 and the summer missions have 5 halo loops. The summer missions require one more halo loop in order to phase the return properly to achieve entry at the proper time of year. This extra halo loop naturally became an enhancement to the mission when Genesis was delayed from a winter to a summer launch.

The trajectory design process and software used is summarized in Figure 12.

The trajectory design process occurs in three stages. First, a baseline trajectory is found using a simple dynamical model. This phase only includes the gravity of the sun, moon, and earth as point masses. Delta-v optimization relative to the LOI maneuver using dynamical systems modeling and differential correction produces an integrated solution that forms the basis for subsequent refinements.

This stage of the design was facilitated through the development of techniques and algorithms developed under Howell at Purdue, and incorporated into and expanded upon in a completely new trajectory design tool at JPL called LTOOL (Libration Point Mission Design Tool).

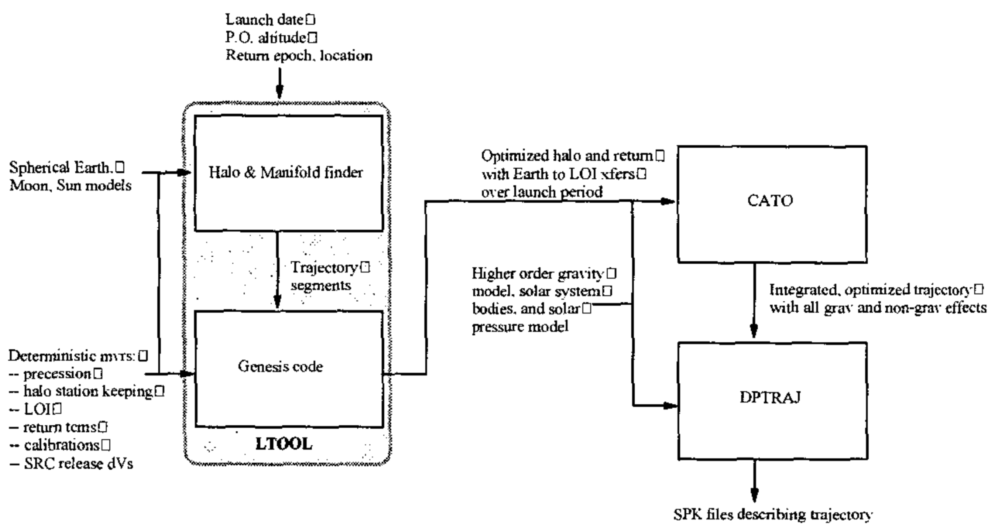

Figure 12. Trajectory Design Process

LTOOL is really a suite of integrated software routines that employ dynamical systems theory in a way that allows the mission design engineer to apply the theory to 3- and 4-body trajectories in a more holistic fashion than other tools available. Though it does not include high-fidelity gravity models or other sophistications that are available in other trajectory tools, the resulting integrated trajectory is precise enough and sufficiently optimized that subsequent enhancements through higher-fidelity modeling and re-optimization with non-gravitational effects and biases require only minor adjustments.

The second stage of the design incorporates more complex gravity models for the three primary bodies, along with the other planets of the solar system. Additionally, the effects of solar pressure and known maneuvers, such as the precession maneuvers required to maintain sun pointing throughout the mission, and calibration maneuvers are added. This is accomplished with an optimization and high-precision trajectory integrating tool called CATO (Computer Algorithm for Trajectory Optimization). The results of CATO are validated independently using another high-precision trajectory propagation program, DPTRAJ (Double Precision Trajectory Program).

The final design step is to incorporate deterministic biases into the trajectory. The biases are needed to meet various spacecraft operational constraints relative to maneuver execution. These biases are added at places where statistical correction maneuver opportunities occur.

Two types of correction maneuvers, required to adjust for orbit determination and maneuver execution accuracy errors, are used on Genesis. Trajectory Correction Maneuvers (TCMs) are used during the transfer and return phases. While in the L1 vicinity, Station Keeping Maneuvers (SKMs) maintain the spacecraft close to the reference halo, or Lissajous, orbit. None of the transfer maneuvers are biased. The SKMs and return leg TCMs all include a 1.0 to 1.5 m/s bias in a roughly sunward direction, as summarized in Table 3.

5.2 Navigating Genesis

The basic principle behind navigating Genesis is to stick as close to the reference trajectory as reasonably possible [9]. This is a different overall methodology than is used on most planetary missions flown by JPL. Generally, JPL is trying to either hit or get close to another planetary body well beyond Earth's gravity. An ideal or reference mission is the basis for navigating those missions as well, but the emphasis is more upon achieving an endpoint condition relative to the target body. The degree of deviation from the reference ideal is of less importance than achieving the proper conditions at the terminus of the mission. That is because the sensitivity of the endpoint objectives to deviations from an interplanetary trajectory have much longer time constants than in the case of a libration point mission like Genesis.

Table 3. Trajectory Biases

Maneuver	Days from Launch	Bias (m/s)	Off-Sun (deg)
SKM-1A	126	1.5	5.1
SKM-1B	161	1.5	5.1
SKM-1C	224	1.5	5.1
SKM-2A	287	1.5	5.1
SKM-2B	350	1.5	5.1
SKM-2C	413	1.5	5.1
SKM-3A	490	1.5	5.1
SKM-3B	546	1.5	5.2
SKM-3C	629	1.5	5.1
SKM-4A	665	1.5	5.1
SKM-4B	721	1.5	5.1
SKM-4C	777	1.5	5.1
SKM-5A	833	1.5	5.1
SKM-5B	889	1.5	5.2
SKM-5C	945	1.5	5.1
TCM-6	988	1.5	7.5
TCM-7	1021	1.5	6.2
TCM-8	1057	1.5	8.5
TCM-9	1097	1	9.5
TCM-10	1117	1	9.7
TCM-11	1126	1	18.5

Libration point missions can usually rely on maintaining relatively loosely controlled trajectories relative to the reference. But the return to Earth imposes the need for much tighter control, since deviating from the reference can quickly lead down a "slippery slope" incurring increasing amounts of ΔV in order to recover the mission. Thus, Genesis "tags up" fairly frequently with the reference trajectory, by placing TCMs and SKMs at regular intervals along it. Targeting for most of the mission is from one maneuver to the next. The exceptions are the first four TCMs, from launch through LOI, and the last three, which are used for terminal entry targeting. The maneuvers are summarized in Table 4.

The numbering system for the SKMs is based on which of the five halo orbits it is in, with an a, b, or c appended, for each of the three SKMs per halo. The first SKM(1a) is really a cleanup maneuver after LOI, completing the insertion into the unstable manifold that brings us home.

5.2.1 Transfer Phase – Early Maneuver Strategy

The most challenging maneuver to design was TCM-1, principally because it occurs so soon after launch, at about 48 hr. Low-energy, non-escaping missions such as Genesis require an accurate injection. This is because any errors, particularly energy dispersions, quickly are magnified as the spacecraft climbs up through the earth's gravity well. Thus, launch errors must be corrected early before they grow unacceptably large. Solid motor upper stages, such as the Star 37FM used on the Delta 7326 launch vehicle, tend to have fairly significant injection errors because they are "point and shoot" spinners that deliver a fixed impulse and do not have guidance systems that provide controlled steering to a guided cutoff, as is the case with most liquid upper stages. Understandably, the Genesis project suffered a certain degree of anxiety over this maneuver, its potential urgency, especially in the case of larger though unlikely errors at launch, and the operational challenges of being ready to perform a major maneuver with a brand new spacecraft. This anxiety was considerably lessened when the actual launch injection proved to be very accurate.

Table 4. Maneuver Summary by Mission Phase

Phase	Propulsive Maneuvers	Purpose
Launch / Transfer	TCMs 1-4	Correct launch injection errors and deliver spacecraft to LOI point
Lissajous Orbit Insertion (LOI)	LOI (TCM-5), SKM-1A	Place spacecraft on desired unstable manifold to satisfy mission objectives
Science (Halo)	SKMs 1B-5C (3 per halo orbit)	Maintain halo orbit station keeping to support Science collection and preparation for return to Earth
Return/Entry	TCMs 6-11	Return to Earth and satisfy nominal entry conditions for SRC and planetary protection requirements for spacecraft deboost.

Since TCMs 1-4 need to work in concert to arrive at the targeted LOI position, an early maneuver strategy ("early maneuver" referring to these first maneuvers) was developed that relied primarily on the first and third TCMs, retaining TCMs 2 and 4 as backups. TCM-1 had the task of correcting launch energy. TCM-3 would manage velocity pointing errors at launch and resolve any remaining energy differential. An additional concern was to make the first TCM as operationally simple as possible, both from the standpoint of ground operations and for the nascent spacecraft systems.

One particular concern was for the viability of the star tracker shortly after launch. Experience with previous missions had shown that star trackers could be temperamental at this early stage due, among other things, to the presence of small particles of debris around the spacecraft associated with launch and separation. These particles might reflect sunlight and mimic stars, thus producing ambiguous attitude information from the tracker. Star tracker problems eventually proved to be non-existent in actual flight.

Using the spinning sun sensors for attitude information, instead of the star tracker, is a valid means to support the first TCM. Most of the energy correction would be in either a sunward or anti-sunward direction, given the geometry of the outbound trajectory is more or less sunward. Thus, this critical first maneuver could be performed using minimal spacecraft capabilities which would already have been available to support other post-launch activities, such as attitude maneuvers to adjust the spin rate and position spacecraft relative to the sun properly to illuminate the solar arrays. Further, it was determined that one of three fixed inertial directions, one sunward and two anti-sunward, would suffice for the entire launch period, allowing a "pre-canned" maneuver build to be used for any day of launch. The canned maneuver sequence had only burn time as a variable. All other aspects of the TCM sequence could remain constant with launch date.

Even with this simplified maneuver, there were limitations on the SSS's ability to maintain accurate estimates of the attitude and spin rate, which constrained the how close to the sun the spin axis could be placed. These are the keep out zones (KOZs), described in an earlier section. For TCM-1, on SSS only, the KOZ is significant – about 25°. By judiciously optimizing targeting between TCM-1, 3, to LOI, the effect of the KOZ at launch was mitigated and residual errors in pointing caused by TCM-1's less-than-optimal orientation could easily be managed at TCM-3.

One other issue that had to be addressed for this early maneuver was the potential for rather significant injection errors in the statistical extremes of launch vehicle errors, particularly due to impulse errors from the solid third stage. The possibility of these larger, lower probability errors motivated including provisions for a contingency maneuver at 24 hours. This managed ΔV budget and burn time constraints associated with the 3σ errors at the planned 48-hour epoch. For either a 24 hr or 48 hr maneuver, the same maneuver sequence would be used, adjusting only burn time.

5.2.2 Science Phase – Managing the Halo Orbit

Navigating a mission relative to a point that represents essentially a construct that possesses certain unique, mathematically definable gravitational and dynamical characteristics relative to the Earth and Sun poses an unusual navigational challenge. This is not a traditional navigational undertaking where missions to solar system bodies with known or definable ephemeredes is the norm. Of course, other missions have been successfully flown relative to libration points, but none had to return to earth.

The reference trajectory, then, becomes the basis for navigating a good deal of the mission, rather than the end point conditions at earth, because the reference is uniquely able to achieve the proper earth return conditions via its association with the appropriate manifold. The entry can ultimately be achieved by simply "sticking to the plan", as it were, and keeping the spacecraft in close proximity with the reference mission. This is the primary basis for navigating the halo, since, just as a trail of breadcrumbs can lead one back home, the reference halo, which occupies an unstable manifold that leads back to earth, provides us with the waypoints that are required to achieve our return.

The station keeping maneuvers (SKMs), which maintain the design manifold or halo, are very similar to orbit maintenance maneuvers used in earth (or any other body) orbiting missions to sustain a specific characteristic relative to that body, such as a repeating groundtrack pattern. The SKMs are spaced at roughly 60-day intervals during the 30-month halo phase, three SKMs per halo rev. The locations are denoted by '+' in the trajectory diagram in Figure 9. Each maneuver is usually targeted to the inertial position of the next SKM, as specified by the reference trajectory. Occasionally, in order to achieve certain maneuver characteristics that are more operationally advantageous, targeting may be optimized between the subsequent two SKMs, but always with the goal of returning to the reference at the last maneuver to be optimized. Predictability of the desired maneuver characteristics is the point of the biasing (Table 3), so, unless the reference trajectory is changed, maneuver-to-maneuver targeting is the expectation in all cases.

5.2.3 Return and Entry Phase

Five maneuver opportunities (TCM-6 through 11) are used to control the return trajectory, most of which are targeted in the same point-to-point manner as is used during the halo. But the return must culminate in hitting specific target conditions, in terms of latitude, longitude and flight path angle (FPA), at atmospheric entry, which is defined to occur at 125 km altitude above a reference radius of 6378.14 km. Terminal navigation is intended not only to reach the proper geocentric location to enable recovery, but also to remain within the constraints of the SRC to withstand the stress of entry. The key parameter to control in meeting the SRC constraints is the flight path angle. The FPA error budget is shared between the spacecraft and the navigation system. A summary of entry requirements is shown in Table 5, and the allocation of entry FPA error is shown in Table 6.

Table 5. Genesis Entry Specifications

Velocity	11.04 km/s
Downrange/crossrange error	18 km X 2 km ellipse @ 125 km altitude
Flight Path Angle	-8.0° ±0.08°

Table 6. Genesis Flight Path Angle Error Allocation

Error Source	FPA Allocation (3σ)
Precession to attitude, spin up, SRC release	0.03°
Orbit Determination, Maneuver Execution	0.05°

The key to controlling these entry conditions is attaining a maneuver execution accuracy that is significantly better than the 6% pre-launch requirement. This requirement is sufficient for most of the mission. But for entry, these errors must be improved by better than half. Part of achieving this accuracy is making use of the ΔV resulting from a spin rate change. For the final TCMs, ΔV will be produced by changing the spacecraft spin rate (up and down) in the desired $\Delta \underline{V}$ direction. The key part of accuracy improvement is correlating spin rate change with ΔV through in-flight calibrations. These scheduled calibrations are achieved through specified spin rate changes during the return phase where ΔV is determined by Doppler tracking. The calibrations reduce the maneuver execution errors to 1% - 2%. This allows the final two maneuvers, TCMs 10 and 11, to control entry errors within the required boundaries.

Table 7 summarizes the return TCMs. TCMs 10 and 11 are the crucial terminal navigation maneuvers used to control the target entry interface conditions specified in Table 5.

Table 7. Return Maneuver Summary

TCM	Time to Entry	Targeting	Maneuver Mode
6	-139d	TCM-7	Timed open loop
7	-106d	TCM-8	Timed open loop
8	-70d	TCM-9	Timed open loop
9	-30d	TCM-10	Timed open loop
10	-10d	Entry	Spin rate control
11	-1d	Entry	Spin rate control

Recall that these return maneuvers are biased 1.0 to 1.5 m/s each (Table 3). The bias direction is intended to keep these maneuvers pointed more or less sunward, consistent with an attitude plan meant to minimize orbit perturbing precessions during the last 60 days of the mission. Though the biases make these maneuvers mandatory, they also keep them predictable in direction and magnitude, enhancing operational simplicity, keeping the solar arrays sun-pointed during the maneuvers, and permitting unbroken telecommunications via the forward LGA without having to consider fore-aft antenna switches which would occur during any maneuver requiring an anti-sunward attitude.

The result of Monte Carlo simulations of these final maneuvers assuming 1% maneuver execution accuracies produces very encouraging preliminary results for the final targeting statistics. Plotted against the allowable 2 X 18 km error ellipse, the results are shown in Figure 13.

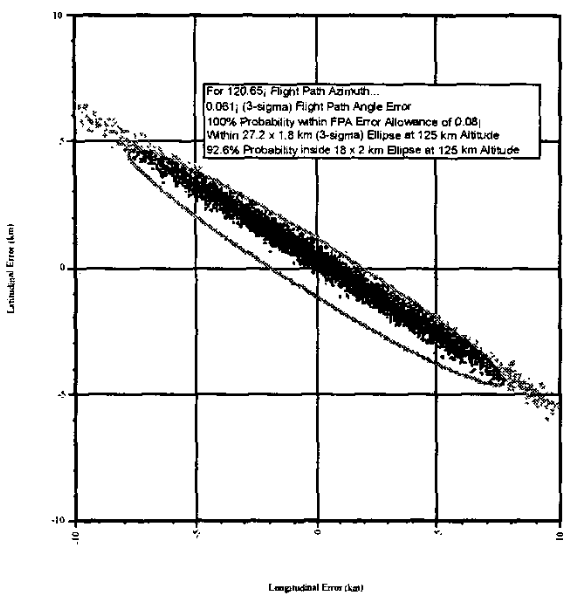

Figure 13. Current Estimate of Entry Performance

Though the results imply only about a 90% confidence of being inside the ellipse, this includes rather pessimistic orbit determination accuracies. Flight experience should improve on those accuracies, allowing us to ultimately achieve a 95% or better capability by the time we approach entry. At any rate, the more critical FPA error of 0.08° is easily met, as indicated in the diagram by the notation that the 3σ level of this error remained within 0.061°. This includes spacecraft separation effects along with all navigation error contributions.

A further assurance is that the predicted ground footprint dispersions, including descent errors, are all well within the 84 x 30 km 3σ ellipse requirement.

5.2.4 Entry Timeline

The final days of Genesis are exceptionally busy, with constant assessment of the approach conditions through tracking and orbit determination and preparations for the terminal maneuvers required to deliver the payload to its destination in Utah [10]. In addition, provisions have been made for "bail out" opportunities to either divert into a 24-day backup orbit, which would allow for a second entry attempt should any circumstance arise requiring a "wave off and go around" that would prevent the planned entry, or, in the worst possible circumstance, to ditch the entire spacecraft into the Pacific Ocean, should some catastrophic development completely rule out the possibility of a safe entry. This latter option is primarily to address public safety issues, which also is the principle motivation behind the requirement to deorbit the spacecraft bus off the northwest US coast following SRC separation, even if entry occurs as planned. All these elements must be rolled up into a timeline of events and potential events, illustrated in Figure 14.

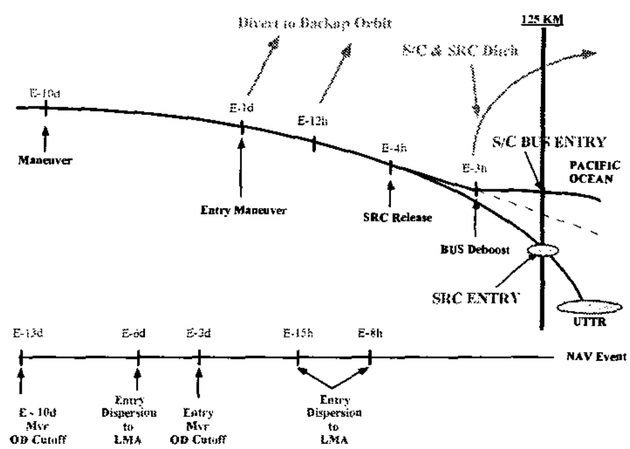

Figure 14. Entry Timeline

The "trajectory", so to speak, of normally expected events follows the black line down through "SRC entry" to UTTR. Significant navigation events are delineated along the timeline at the bottom. Red is used to denote where contingency events would occur. Note there are two opportunities, at –24 hr and –12 hr, to divert to the backup orbit. Obviously, for the "ditch" option, the SRC release at –4 hr would not have taken place. In fact, a failure of the SRC to release would be one reason to execute the ditch maneuver. In any case, whether the spacecraft is on the nominal plan and the SRC is on its way to UTTR, or a significant anomaly forces the "ditch" scenario, the same pre-programmed maneuver will be executed by the bus.

5.2.5 Backup Orbit

If a nominal direct return becomes impossible, two maneuvers to the backup orbit are required. The first, about 10 m/s at 12 hours from entry (if we wait until the latest opportunity for this diversion), raises the perigee of the incoming trajectory above the atmosphere, to about 200 km. The second, a retrograde maneuver of about 30 m/s, places the spacecraft in a 24-day elliptical orbit (Figure 15).

An entry targeting maneuver of 65 m/s at apogee 12 days later reduces perigee to permit atmospheric entry at the subsequent perigee passage on October 2, 2004. If deemed necessary, the apogee maneuver could be delayed until the second apogee passage, and the backup entry would occur another 24 days later, on October 26. Options for delaying the entry even later are not attractive because of the possibility of inclement weather as winter approaches in the Utah recovery area.

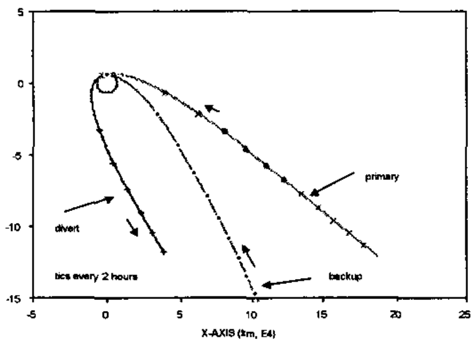

Figure 15. Backup Orbit

6. Systems Integration and Test

Systems integration and test was a key part of the Genesis development effort. This work included delivery of tested subsystems, integration of those subsystems into an overall flight system, system functional and performance testing, mission compatibility testing, system level environmental testing, spin balance testing, system level stress testing and operations readiness testing.

Subsystems were tested and delivered to the overall system. These tests were completed to insure integrity and verify compliance with subsystem level requirements. These subsystems were then integrated into an overall flight system. Functional testing was performed to verify proper interfaces between subsystems. These subsystems included propulsion, structures, mechanisms, thermal, ACS, telecommunications, C&DH, EPS, and flight software.

Upon integration into an overall flight system, system performance testing was completed. Successful ACS sensor and phasing tests were essential to preparing the flight system for launch. System level testing of mission scenarios was also performed to demonstrate compatibility between the flight system and mission operations products, e.g. mission blocks and sequences.

System level environmental testing was performed to verify compatibility of the flight system and the defined mission environments. These tests were key to demonstrating spacecraft function in the most flight-like environments possible. System level environmental testing included acoustics, pyro shock, thermal vacuum, and EMI/EMC testing. Figure 16 shows the flight system in the thermal vacuum configuration.

Additional testing in preparation for launch included spin balance testing, system level stress testing, and operations readiness testing. Spin balance was especially important to Genesis to demonstrate mass properties with acceptable spin stability across various mission configurations, including launch, cruise, science, and return. Figure 17 displays the flight system in the spin balance configuration. System level stress testing was performed to validate performance across a range of flight conditions. Finally, operations readiness testing was performed to train the mission operations team under simulated flight conditions and to certify the various maneuver types to be implemented in flight.

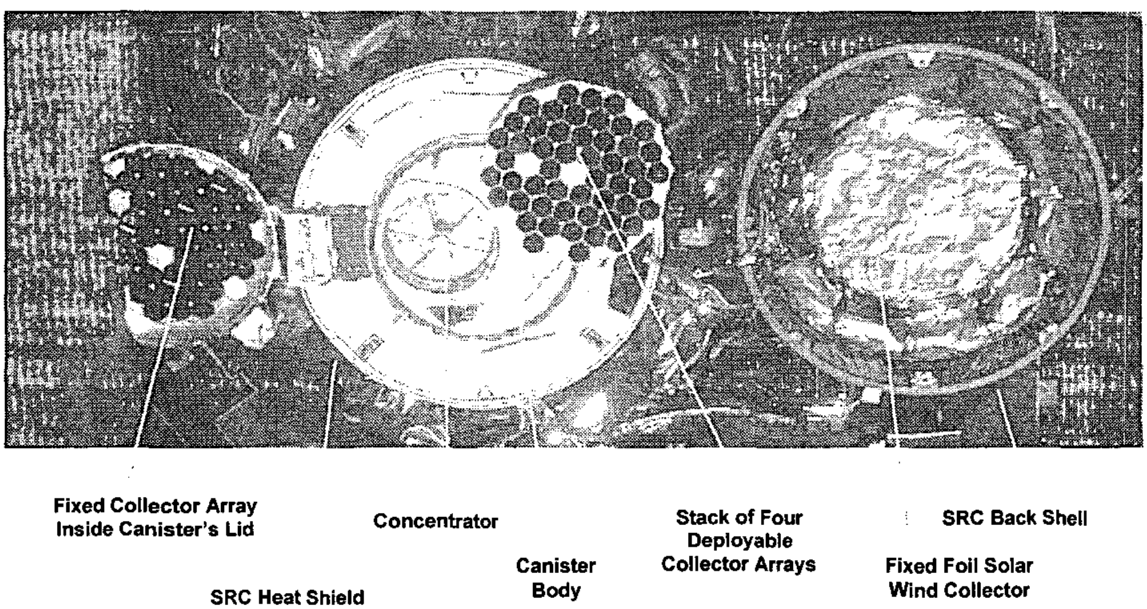

Figure 16. Flight System Thermal Vacuum Configuration

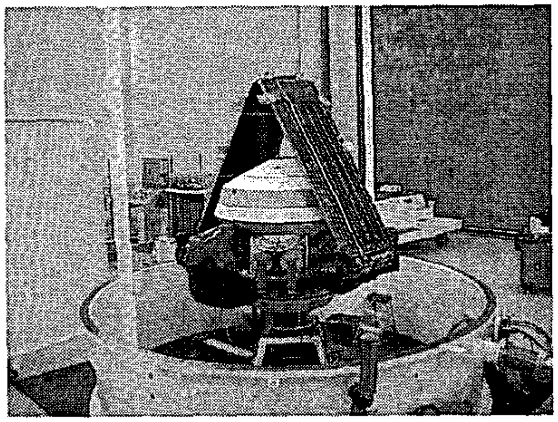

Figure 17. Flight System Spin Balance Configuration

Systems integration and test played an essential role in preparing Genesis for flight. This testing verified the readiness of the flight system and the mission operations team. It also demonstrated the compatibility between the flight system and the mission operations approach.

7. Mission Operations

Mission operations planning has been included from the beginning of Genesis design and development. The PCAR (Planning, Control, Analysis and Recovery) organization helped establish mission requirements, especially those related to flight operations, and coordinated mission design and navigation planning. PCAR also was responsible for the Operational Readiness Testing (ORT) that performed end to end testing from activity planning through ground and spacecraft execution.

The mission operations function is divided into three teams: spacecraft, planning and navigation, and science. The spacecraft team (SCT), located in Denver, plans detailed spacecraft activities, generates command sequence blocks, tests and uplinks sequences, monitors engineering telemetry, and trends subsystem performance. The planning and navigation team at JPL plans mission activities, including trajectory corrections and orbit determination, coordinates with the Deep Space Network (DSN) for spacecraft tracking and communications, and integrates command sequences. The science team, whose members are at Caltech, JPL and Los Alamos, monitors science operations and analyzes science return.

The mission and spacecraft design process co-evolved with mission operations design. Mission timelines and command blocks were established in parallel with spacecraft design and test, including solar array and battery electrical power timelines (loads vs. capabilities) and ACS component performance analyses. Other hardware, software and operations concurrent development established flight rules and constraints, and procedures for transforming navigation maneuver requirements through maneuver decomposition into spacecraft commands.

Mission planning and resources (work schedules, spacecraft tracking, simulation laboratory testing, etc.) all were integrated and de-conflicted with a large armada of active missions. For the SCT, that included Genesis, Stardust, Mars Odyssey and Mars Global Surveyor. For JPL, in addition to these missions, there are Galileo, Ulysses, Deep Space 1, and several other missions. All missions 'compete' for the resources of the tracking and operations network.

8. Operational Status

Genesis was launched from Cape Canaveral on August 8, 2001 after a 9 day delay related to a combination of weather and star tracker power converter concerns (later determined to be not applicable). The launch and injection by the Boeing Delta 7326 was almost flawless. The first scheduled midcourse maneuver, at 48 hours after spacecraft separation from the launch vehicle, required 5 m/s delta-V, just at the minimum threshold established pre-launch for doing any maneuver at all. No other correction maneuvers were needed before LOI.

All subsystems performed well within their pre-flight predicted performance. Solar arrays were deployed within a minute after spacecraft separation, and precession to near sun pointing occurred shortly afterward. The star tracker was turned on, checked out, and is performing very well. The SRC back shell was opened for outgassing at 11 days after launch, as scheduled. All activities and performance were according to plan, except the SRC component temperatures.

After the SRC was opened, component temperatures started rising more quickly than anticipated. Of immediate concern was the SRC battery, where thermal predictions during science collection would result in exceeding the flight allowable maximum temperatures. Excessive battery heating may compromise its power load capacity, and possibly jeopardize release of the parafoil during descent over Utah.

The most likely cause of increased temperatures is contamination of SRC internal painted surfaces, including the battery radiator. The contaminants, when exposed to ultraviolet light from the sun, may have polymerized on the paint surface to change its thermal properties. Testing and analyses of paint samples, contamination sources, engineering science canister and flight like battery cells, have been, and continue to be, performed. It is expected that revised flight allowable temperatures will not be exceeded.

The science collection mission has been started after a successful LOI. Quality science collection time is expected to greatly exceed the minimum 22 month requirement. Other performance measures have also exceeded expectations, such as a projected fuel use of only 25% of the loaded fuel, and nutation damping that is about 30% faster than predicted, allowing more time for science collection. The responsiveness and robustness of the operations team, excellent spacecraft health and high solar variability all point toward achieving a very successful mission.

9. CONCLUSIONS AND LESSONS LEARNED

The path from Genesis proposal to flight operations has been an evolutionary development. Like many programs, unexpected challenges presented obstacles that threatened Genesis' successful development and operations. Yet, the entire team was able to create integrated innovative solutions that not only resolved potential concerns, but actually improved the mission. The lessons learned include the following.

- **Practical experience and insight are invaluable, so is 'out of the box' thinking.** The use of dynamical systems theory, or non-linear manifolds in phase space, enabled the design of a Genesis trajectory that moves from launch vehicle injection through five loops around L1, a trip around L2, and a daylight descent over the recovery site in Utah. This could be done with only one small deterministic delta-V, not counting navigation and biasing effects, for the entire 3 year mission.
- **Look with suspicion at any performance deviation from analyses or test, and be aware of its system and mission impacts.** The potential of degraded star tracker performance during development, and the lack of an IMU, resulted in the development of the spin track attitude estimation algorithm. It also forced the employment of larger sun related keep out zones and the design of operational maneuvers to avoid them.
- **Maintain organizational communication, vertically and horizontally: Help may come from any source.** The close coordination between Lockheed Martin spacecraft ACS/Systems and JPL MDNav resulted in the development, and use, of maneuver decomposition and maneuver types to produce the desired net delta-V with the appropriate spacecraft configuration and constraints. A maneuver type certification process was instituted to make sure all interfaces and tools were valid from an end-to-end perspective.
- **Don't be afraid to draw from techniques and processes normally reserved for operational contingencies as long as adequate performance margin can be preserved.** The use of trajectory biasing and minimum delta-V thresholds maximized the probability that midcourse corrections would be relatively benign operationally, rather than involving complex activities that increased mission risk.
- **Incorporate operations development concurrently with spacecraft and mission development.** The operational concept of changing the spin rate to produce delta-V, and using in-flight calibrations to reduce execution error, was introduced early in Genesis development. This mode of delivering delta-V is critical for the final midcourse corrections to ensure an accurate Earth entry.
- **Test, test, test:** A wide range of testing is mandatory, such as test as you fly, operational readiness tests, risk reduction testing. An example of a testing benefit was the simulation of a failure scenario that resulted in determining the correct sequence of autonomous recovery activities, and a practical value for the level of battery state to trigger a fault protection response.

Whether the program philosophy is Faster, Better, Cheaper or Mega-Mission, it is not sufficient to be programmatically and technically 'excellent', and to maximize mission success probability. Space missions are so complex that some things occasionally go wrong. Risk reduction testing and Monte Carlo simulation analyses can reveal unusual and subtle failure circumstances, but are only as good as test environments and models.

A lot more can be learned from mistakes than from successes. It's related to the 'couldn't imagine' syndrome. We couldn't imagine that Velcro would cause a spark that would ignite the oxygen rich environment of Apollo 1. We couldn't imagine that a minor units error could ripple through the layers of design, test and operations to cause MCO to burn up in the Martian atmosphere. We couldn't imagine that a group of men would take over a commercial airliner and crash it into a skyscraper. We cannot imagine how many catastrophic scenarios have come close to realization, and escaped our awareness.

Some life lessons. Innovation and paradigm shifts are created by pushing the envelope of our imaginations, but are actualized by real-world experience. The skill and motivation of the project team stacks the odds heavily in favor of mission success. Yet, the practical limitations of available resources and the ever present unknown, mean there is no sure thing. Ironically, it is sometimes as important to be lucky as it is to be good.

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BIOGRAPHIES

Phil Hong has a B.A. in Physics and an M.S. in Aerospace Engineering. He has been involved in mission, navigation and systems analysis and design for over 36 years. Phil initially worked on the design and operations of all 5 Lunar Orbiter missions, then Apollo 8 through 13, and is currently supporting Genesis operations. He has been involved in a wide spectrum of projects including interplanetary and Earth orbital spacecraft, space technology, launch vehicles and strategic systems.

George Carlisle has his degree in Aerospace Engineering from the U of Texas at Austin, and has spent 27 years of his more than three decades in the space exploration field at JPL. Most of that time has been in the Navigation and Mission Design Section. George has worked, mainly performing trajectory analysis and mission design, for a variety of projects, including Voyager, Galileo, Topex/Poseidon, and Deep Space 1. He is currently the Mission Design and Navigation Team Chief for the Genesis mission.

Nick Smith has a B.S. in Math and an M.S. in Aerospace Engineering, with over 17 years of aerospace experience in mission analysis and systems engineering. Flight programs have included the Space Shuttle Solid Booster Parachute System, Mars Global Surveyor, Stardust, and Genesis. Advanced studies have included Lunar Resource Mapper, Solar Probe, Saturn Miniprobe, and Mars Sample Return. Nick is the Chief Engineer for the Genesis spacecraft.

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